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RESEARCH MEMORANDUM

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WIND-TUNNEL INVESTIGATION OF THE AERODYNAMIC CHARACTERISTICS IN PITCH OF WING-FUSELAGE COMBINATIONS AT HIGH SUBSONIC SPEEDS

TAPER-RATIO SERIES

By Thomas J. King, Jr., and Thomas B. Pasteur, Jr.

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WIND-TUNNEL INVESTIGATION OF THE AERODYNAMIC CHARACTERISTICS IN PITCH OF WING-FUSELAGE COMBINATIONS AT HIGH SUBSONIC SPEEDS

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SUMMARY

The results presented in the present paper represent a continuation of a program being conducted in the Langley high-speed 7- by 10-foot tunnel to investigate the aerodynamic characteristics in pitch, sideslip, and steady roll of model configurations having variations in the wing geometric parameters. Presented herein are the aerodynamic characteristics in pitch of wing-fuselage combinations with wings of aspect ratio 4, sweepback angle of 45°, and taper ratios of 0.3, 0.6, and 1.0. The Mach number range was from 0.40 to about 0.95 and the Reynolds number ranged from 2,000,000 to 3,000,000.

The results of the investigation indicate at low lift coefficients a reduction in lift-curve slope and a forward movement in aerodynamic center with an increase in taper ratio throughout the test range of Mach number, as would be predicted from available theory. All wings showed a rapid forward movement in aerodynamic center at the higher lift coefficients; however, the lift coefficient at which this forward movement started was found to increase with increased taper ratio.

Only small differences in minimum drag, drag due to lift, and lift-drag ratios resulted from variation in taper ratio for the constant-thickness-ratio wings investigated. Adjustment of the thickness ratio to provide equal aeroelastic characteristics may allow some improvement in minimum drag and in lift-drag ratios as the taper ratio is reduced, at least at the higher Mach numbers.

INTRODUCTION

A systematic research program is being carried out in the Langley high-speed 7- by 10-foot wind tunnel to determine the aerodynamic characteristics of various model configurations in pitch, sideslip, and during steady rolling up to a Mach number of about 0.95. The Reynolds number range for the sting-supported models varies from 1,500,000 to 6,000,000, depending on the wing plan form and the test Mach number. The Reynolds number for the taper ratio series of wings varied from about 2,000,000 to 3,000,000.

The wing plan forms used in the current research program are similar, in general, to the plan forms investigated at lower Reynolds numbers during a previous research program which utilized the transonic-bump technique for obtaining results at transonic speeds. Some of the results obtained from the transonic-bump program have been summarized in reference 1. Some similar or related wing plan forms also have been investigated in other facilities. (For examples, see refs. 2 and 3). A comparison of aerodynamic characteristics in pitch, as obtained by different test techniques, has been reported in reference 4. All wings of the present program have the NACA 65A006 airfoil section parallel to the plane of symmetry. As previous parts of the program, the effects of aspect ratio on the pitch characteristics of 45° sweptback wings of taper ratio 0.6, and the effects of sweep angle on the pitch characteristics of a series of wings having aspect ratio 4 and taper ratio 0.6 are presented in references 5 and 6, respectively.

The present paper presents results of an investigation of the effects of taper ratio on the aerodynamic characteristics in pitch of 45° sweptback wings of aspect ratio 4 when mounted on the same fuselage used for other parts of the program.

COEFFICIENTS AND SYMBOLS

The symbols used in the present paper are defined in the following list. The forces and moments are referred to a wind-axes system with origin located at the quarter-chord point of the mean aerodynamic chord.

C_L lift coefficient, $\frac{\text{Lift}}{qS}$

C_D drag coefficient, $\frac{\text{Drag}}{qS}$

C_m	pitching-moment coefficient, $\frac{\text{Pitching moment}}{qS\bar{c}}$
ΔC_D	drag due to lift, $C_D - C_{D_{C_L=0}}$
q	dynamic pressure, $\frac{1}{2}\rho V^2$, lb/sq ft
S	wing area, sq ft
$\bar{c}$	mean aerodynamic chord, $\frac{2}{S}\int_0^{b/2} c^2\,dy$, ft
c	local wing chord, ft
b	wing span, ft
ρ	air density, slugs/cu ft
V	free-stream velocity, ft/sec
R	Reynolds number of wing based on $\bar{c}$, and evaluated in accordance with reference 7
M	Mach number
α	angle of attack, deg
$\Delta\alpha$	local angle-of-attack change due to the distortion of wing
K	correction factor for C_{L_α} due to wing distortion
C_{L_α}	lift-curve slope per degree, $\partial C_L/\partial\alpha$
$\Delta\left(\frac{\partial C_m}{\partial C_L}\right)$	incremental change in aerodynamic-center location due to wing distortion, fraction of mean aerodynamic chord
y	spanwise distance from plane of symmetry, ft
λ	taper ratio

Subscripts:

F fuselage alone

WF wing-fuselage combination

MODELS AND APPARATUS

The wing-fuselage combinations tested are shown in figure 1. All wings had an NACA 65A006 airfoil section parallel to the plane of symmetry and were attached to the fuselage in a midwing position. The wings of taper ratio 0.3 and 1.0 were constructed of solid aluminum alloy and the wing of taper ratio 0.6 was of composite construction, consisting of a steel core and a bismuth-tin covering. The aluminum fuselage used in the present investigation was the same as used for those investigations reported in references 5 and 6 and is defined by the ordinates presented in table I.

The wing-designation system, described in reference 5, has been applied to the present series of wings. For example, the wing designated by 45-4-0.6-006 has the quarter-chord line swept back 45°, an aspect ratio of 4, and a taper ratio of 0.6. The number 006 refers to the airfoil designation - in this case the design lift coefficient is zero and the thickness is 6 percent of the chord.

The models were tested on the sting-type support system shown in figure 2 which has provision for a remotely controlled variation in angle of attack over a range of 28°. The internally mounted strain-gage balance used to measure wing-fuselage forces and moments is shown installed in a wing-fuselage combination in figure 3.

TESTS AND CORRECTIONS

The tests were conducted in the Langley high-speed 7- by 10-foot wind tunnel through a Mach number range from 0.40 to 0.95. Measurements of lift, drag, and pitching moment were made through an angle-of-attack range from -2° to 26°, except when more stringent limitations were imposed because of the available wind-tunnel power, balance capacity, or model strength. The size of the models caused the tunnel to choke at Mach numbers of about 0.94 or 0.95 for the zero-lift condition.

Blocking corrections, which were applied to the Mach numbers and dynamic pressure, were determined by the method of reference 8. Jet-boundary corrections, applied to the lift and drag, were calculated by the method of reference 9. The jet-boundary correction to pitching moment was considered negligible.

No tare corrections were obtained; however, previous experience (ref. 10, for example) indicates that for tailless sting-mounted models of the type investigated herein, the tare corrections to lift and pitching moment are negligible. The drag data have been corrected by an increment obtained by adjusting the pressure at the base of the fuselage to equal the free-stream static pressure. For this correction, the base pressure was determined by measuring the pressure inside the fuselage at a point about 9 inches forward of the base. The resulting drag corrections, which were added to the measured drag coefficients, varied from 0.001 to 0.004 for the three wing-fuselage combinations and from 0.001 to 0.002 for the fuselage alone as the Mach number was increased from 0.40 to 0.95.

The test wings were known to deflect under load. Accordingly, in an effort to correct the measured data to the rigid case, correction factors for the effects of aeroelastic distortion were determined. In order to represent the distortion of the wing in an approximate manner, an elliptic load distribution was simulated by applying loads at four spanwise locations along the quarter-chord line of each wing. The resulting spanwise variation in angle of attack $\Delta\alpha$ was measured (fig. 4) and strip theory was used to calculate the effect of this angle-of-attack variation on the lift and lift distribution from which the correction factors of figure 5 were determined. Results from independent calculations, using beam theory and including the effects of aeroelastic distortion on the span load distribution, are in good agreement with the results obtained by this analysis.

The variations with Mach number of the mean test Reynolds number for the wings tested are presented in figure 6. The Reynolds numbers given in figure 6 were evaluated by using the charts and formulas of reference 7, and are somewhat smaller in magnitude than the values indicated in references 5 and 6. The difference in magnitudes can be attributed to a difference in the method for evaluating the influence of temperature on the viscosity of air, and in this sense the method used to determine the values of Reynolds number presented herein is regarded as being the more accurate.

RESULTS AND DISCUSSION

The basic data for the wing-fuselage combinations having wings of taper ratio 0.3 and 1.0 are presented in figures 7 and 8, respectively. The basic data for the taper-ratio-0.6 wing and for the fuselage alone are presented in reference 5. None of the basic data have been corrected for the effects of aeroelastic distortion. Summary plots of some significant aerodynamic parameters at zero lift (with corrections for aeroelastic distortion applied) are presented in figures 9 to 16. Some additional comparisons of aerodynamic characteristics through the lift range are shown in figures 17 to 20.

Lift Characteristics

The experimental lift-curve slopes measured near zero lift (with and without the aeroelasticity correction applied) are compared with rigid-model theory in figures 9 to 11. The theoretical results were evaluated by the same method used in reference 6; that is, the increment of C_{L_α} at zero Mach number due to the fuselage and wing-fuselage interference was evaluated from the wing-fuselage theory of reference 11 and this increment was applied to the wing-alone theory of reference 12 throughout the Mach number range, as follows:

$$\left(C_{L_{\alpha_{WF}}}\right)_M = \left(C_{L_{\alpha_W}}\right)_M + \left[\left(C_{L_{\alpha_{WF}}}\right)_{M=0} - \left(C_{L_{\alpha_W}}\right)_{M=0}\right]$$

For the wings of taper ratio 0.3 and 0.6, the predictions obtained by this method are in good agreement with experiment except at Mach numbers above about 0.8, where the predicted effects of compressibility are somewhat too small. Similar results have been noted previously. (For example, see ref. 13.) The rather poor agreement between predictions and experiment for the taper ratio 1.0 wing seems to result from inaccuracy of the method at zero Mach number.

A comparison of the experimental lift-curve slopes for the three wings (fig. 12) indicates, as would be expected, a consistent increase in C_{L_α} with decrease in taper ratio throughout the Mach number range. Experimental and predicted results are presented as functions of taper ratio in figure 13. In general, the agreement is good at a taper ratio of 0.3, and, since the predicted variation with taper ratio is larger than that obtained experimentally, the largest discrepancies occur at the highest taper ratio ($\lambda = 1.0$).

Pitching-Moment Characteristics

The slopes of the pitching-moment curves $\left(\partial C_m/\partial C_L \text{ at zero lift}\right)$ with and without corrections for aeroelastic distortion are compared with predictions based on rigid-wing theory in figures 9 to 12. The predicted results were obtained by modifying the wing-alone theory by the same procedure indicated previously for lift-curve slope.

The experimental values of $\partial C_m/\partial C_L$ for these wings show gradual rearward shifts in aerodynamic center with increase of Mach number to 0.85 with small variations for the taper ratio 0.6 and 1.0 wings but relatively large variations for the 0.3 tapered wing. In the range of Mach number from 0.85 to 0.95 (the highest value attained), large rearward shifts of the aerodynamic center occurred. Although the experimental and predicted values of $\partial C_m/\partial C_L$ are in agreement at $M = 0.6$, the predicted values show essentially no variation over the Mach number range for which they are considered applicable. The agreement between experimental and predicted values of $\partial C_m/\partial C_L$ below a Mach number of 0.9 is somewhat better for the wings of taper ratio 0.6 and 1.0 than has been indicated for the wing of taper ratio 0.3. A comparison of curves of $\partial C_m/\partial C_L$ plotted against Mach number for the three wings is shown in figure 12.

Comparisons of experimental and predicted variations of $\partial C_m/\partial C_L$ with taper ratio for certain selected Mach numbers are shown in figure 14. Both experiment and theory indicate a forward shift in aerodynamic center with increasing taper ratio, and the agreement is reasonably good for Mach numbers at least as high as 0.9.

A comparison of curves of C_m plotted against C_L for the three wings under investigation is presented in figure 17 for four selected Mach numbers. In order to provide a fairly realistic basis for comparison of high-lift pitching-moment characteristics, the assumed center-of-gravity locations for the wings of taper ratio 0.3 and 1.0 were adjusted to give the value of $\partial C_m/\partial C_L$ at $C_L = 0$ and at $M = 0.6$ that had been obtained for the wing of taper ratio 0.6. The comparison shows that all wings have a pitch-up tendency (large forward shift in aerodynamic center) at the higher lift coefficients. The wings differ, however, in the lift coefficient at which pitch-up occurs and in the character of the curves before pitch-up. In general, all wings show some tendency toward increasing stability prior to pitch-up, and this increase in stability is more abrupt for the wings having the higher taper ratios. The pitch-up tendency or forward shift in aerodynamic center occurs at higher lift coefficients as the taper ratio is increased. This fact probably can be attributed to the smaller section lift coefficients at the wing tips and, consequently, a reduced tendency to tip stalling for the wings having the larger tip chords.

Drag Characteristics

Drag at zero lift.- A comparison of the zero-lift drag for the three wing-fuselage combinations is presented in figure 12. The lowest drag was obtained for the taper-ratio-0.3 wing; however, the differences in drag for the three wings were very small throughout the Mach number range investigated. Figure 15 gives the zero-lift drag for the fuselage alone, based on wing area. Data for wing plus wing-fuselage interference drag are obtained by subtracting the fuselage-alone drag of figure 15 from the wing-fuselage drag of figure 12.

Drag due to lift.- Drag-due-to-lift characteristics for the three wings are compared as figures 18 and 19. Although the differences are generally small, the highest drag-due-to-lift values are obtained consistently (at least at lift coefficients above 0.4) with the taper-ratio-0.3 wing. At lift coefficients below 0.65, all wings show reductions in drag due to lift with increased Mach number (fig. 19).

Lift-Drag Ratios

The highest values of maximum lift-drag ratio were obtained with the taper-ratio-1.0 wing, except possibly at Mach numbers above 0.80. The differences in values of $(L/D)_{max}$ for the three wings, however, are very small and are probably of little significance. All three wings show an abrupt reduction in $(L/D)_{max}$ at Mach numbers above about 0.91.

Lift-drag ratios are plotted as a function of lift coefficient at four selected Mach numbers in figure 20. As was pointed out with regard to $(L/D)_{max}$, the effect of taper ratio on L/D throughout the lift-coefficient range is quite small. Some superiority of the taper ratio 1.0 wing is again shown at high lift coefficients and at Mach numbers of 0.91 and 0.93.

In comparing the performance characteristics of the particular series of wings under investigation, the fact should be borne in mind that the ratio of wing-section thickness to chord was maintained constant at 0.06 for the three wings. An indication of the effect of taper ratio on the aeroelastic distortion characteristics of the three wings can be obtained by comparing the curves of $\Delta\alpha/qC_L$ given in figure 4 for the wings having taper ratios of 0.3 and 1.0. (The taper-ratio-0.6 wing should not be included in this comparison because the materials used in its construction were not the same as those of the other two wings.) The angular distortion for the taper-ratio-0.3 wing is only about 60 percent as large as that of the taper-ratio-1.0 wing. It is evident, therefore, that, for the same aeroelastic properties, the thickness-chord ratio could be reduced somewhat as the taper ratio is decreased, and this in turn would be expected to result in improved performance characteristics for the wings of lower taper ratio - at least in the higher range of Mach numbers.

CONCLUSIONS

The results of the investigation at high subsonic speeds of a series of wings of varying taper ratio and with an aspect ratio of 4, a quarter-chord sweepback angle of 45°, and an NACA 65A006 airfoil section indicate the following conclusions:

1. The lift-curve slope decreased with an increase in taper ratio throughout the test range of Mach numbers, as would be predicted by available theory.

2. The aerodynamic center at low lift coefficients moved forward with an increase in taper ratio at all test Mach numbers, as indicated by theory. All wings showed a rapid rearward movement of aerodynamic center above about 0.85 Mach number; however, only the taper-ratio 0.3 wing showed an appreciable rearward shift within the lower Mach number range.

3. All wings showed a rapid forward movement in aerodynamic center at the higher lift coefficients; however, the lift coefficient at which this forward movement started was found to increase with increased taper ratio.

4. For the series of wings investigated, in which the ratio of section thickness to chord was maintained constant, there were only very small differences in minimum drag, drag due to lift, or lift-drag ratios for the various wings. The aeroelastic distortion was reduced, however, as the taper ratio was reduced, and therefore, if the thickness ratios had been adjusted to provide more nearly equal aeroelastic characteristics for the three wings, it is possible that some improvement in minimum drag and in lift-drag ratios would have resulted from a reduction in taper ratio, at least for the higher Mach numbers.

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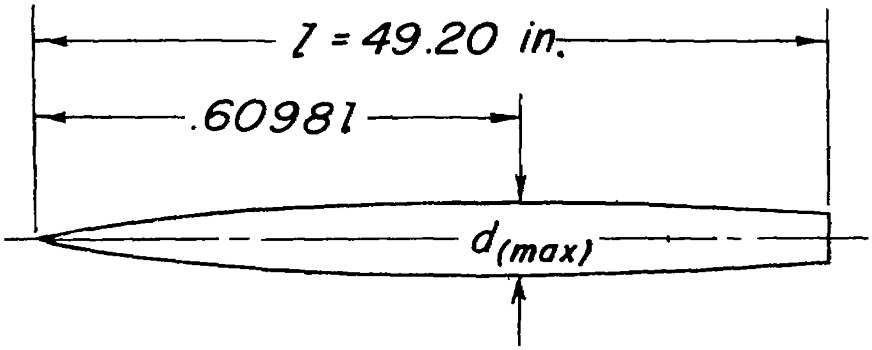
TABLE I.- FUSELAGE ORDINATES

[Basic fineness ratio 12, actual fineness ratio 9.8 achieved by cutting off rear portion of body]



Ordinates, percent length	
Station	Radius
0	0
.61	.28
.91	.36
1.52	.52
3.05	.88
6.10	1.47
9.15	1.97
12.20	2.40
18.29	3.16
24.39	3.77
30.49	4.23
36.59	4.56
42.68	4.80
48.78	4.95
54.88	5.05
60.98	5.08
67.07	5.04
73.17	4.91
79.27	4.69
85.37	4.34
91.46	3.81
100.00	3.35
L. E. radius = .0006l	

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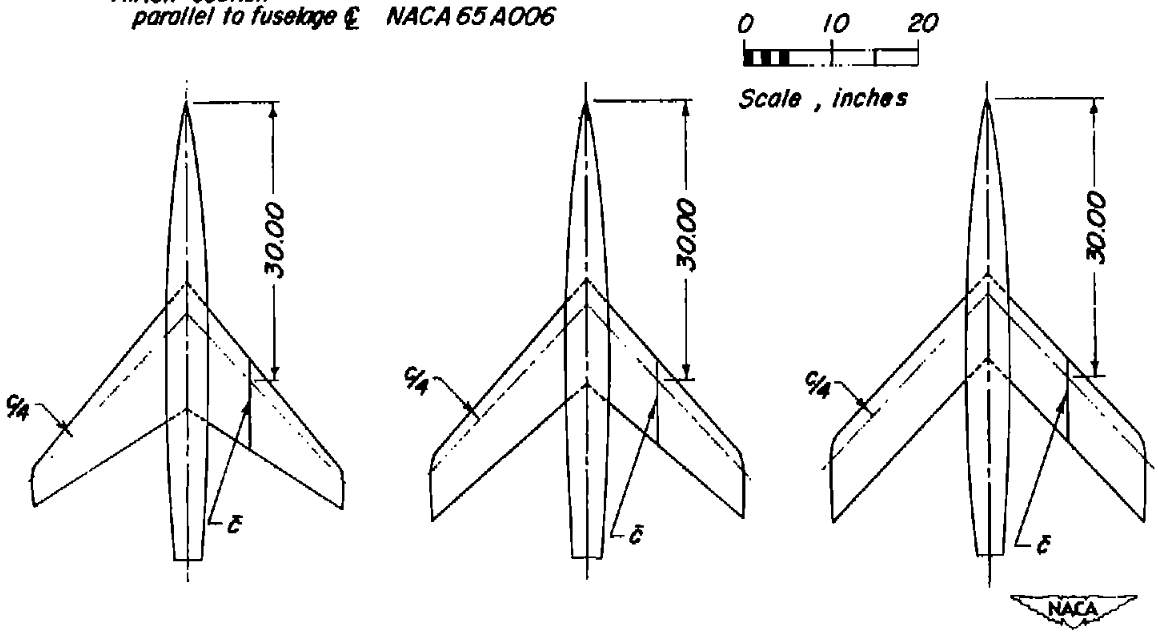
Wing geometry

Area	2.25 sq ft
Span	3.00 ft
Sweep at 0.25 chord line	45°
Aspect ratio	4
Incidence	0
Dihedral	0
Airfoil section parallel to fuselage ℄	NACA 65A006

Fuselage

Length	49.20 in.
Max diam.	5.00 in.
Position of max diam. (from nose of model)	30.00 in.



Wing 45-4-0.3-006

Taper ratio	0.3
Root chord	13.85 in.
Tip chord	4.16 in.
Mean aerodynamic chord	0.822 ft

Wing 45-4-0.6-006

Taper ratio	0.6
Root chord	11.25 in.
Tip chord	6.75 in.
Mean aerodynamic chord	0.765 ft

(Basic data presented in Ref. 5)

Wing 45-4-1.0-006

Taper ratio	1.0
Root chord	9.00 in.
Tip chord	9.00 in.
Mean aerodynamic chord	0.750 ft

Figure 1.- Drawing of the three wing-fuselage configurations.

Figure 2.- Model installed on the variable-angle sting support used in the Langley high-speed 7- by 10-foot tunnel.

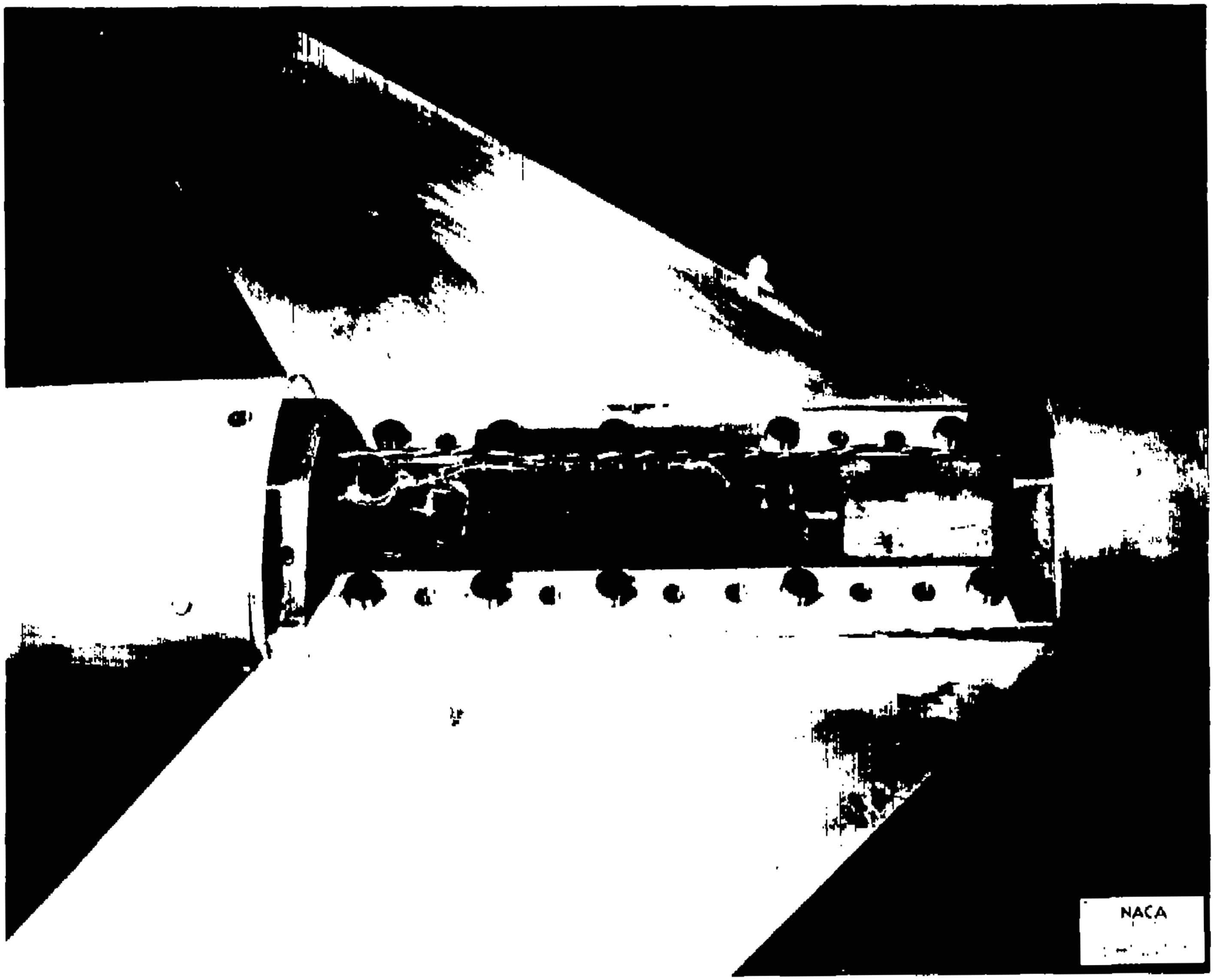


Figure 3.- Model showing installation of the strain-gage balance.

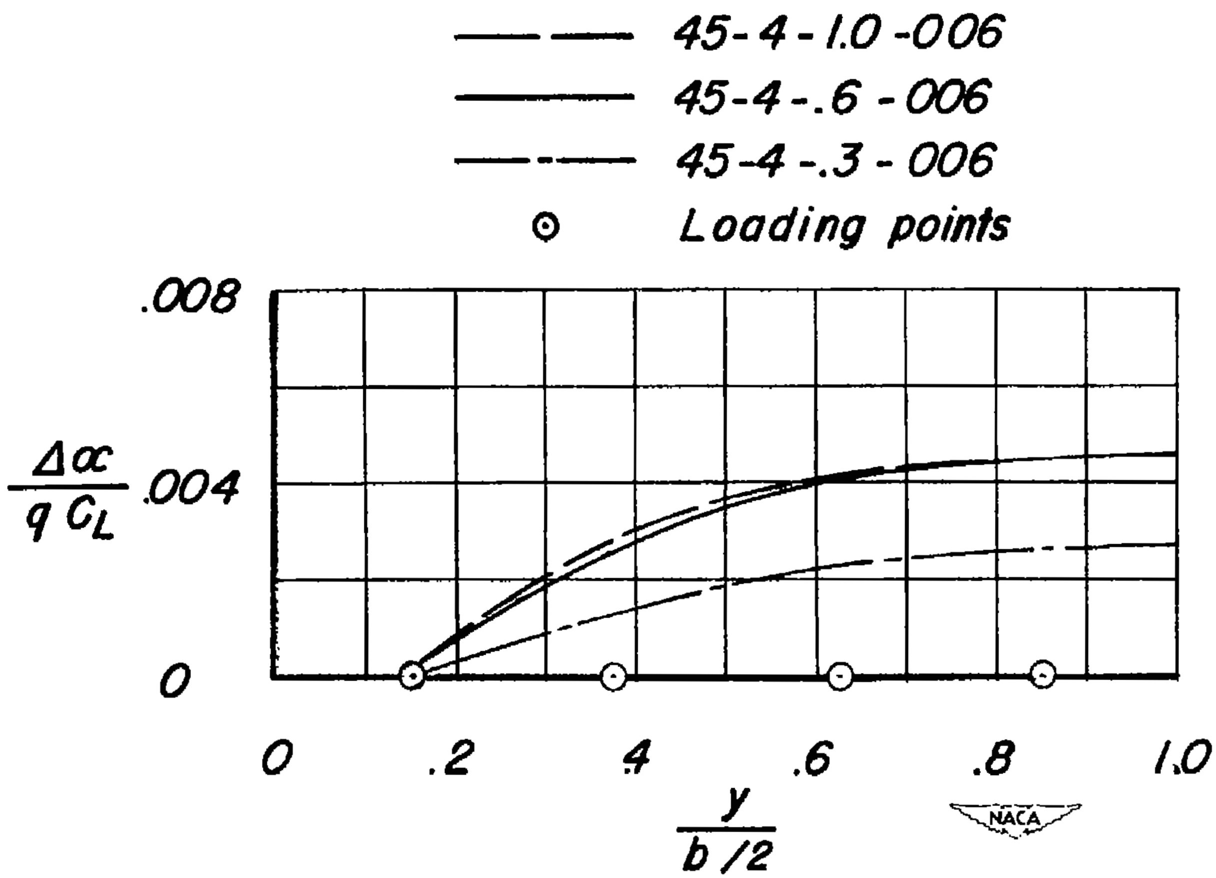


Figure 4.- Spanwise variation of angle of attack due to aeroelastic distortion.

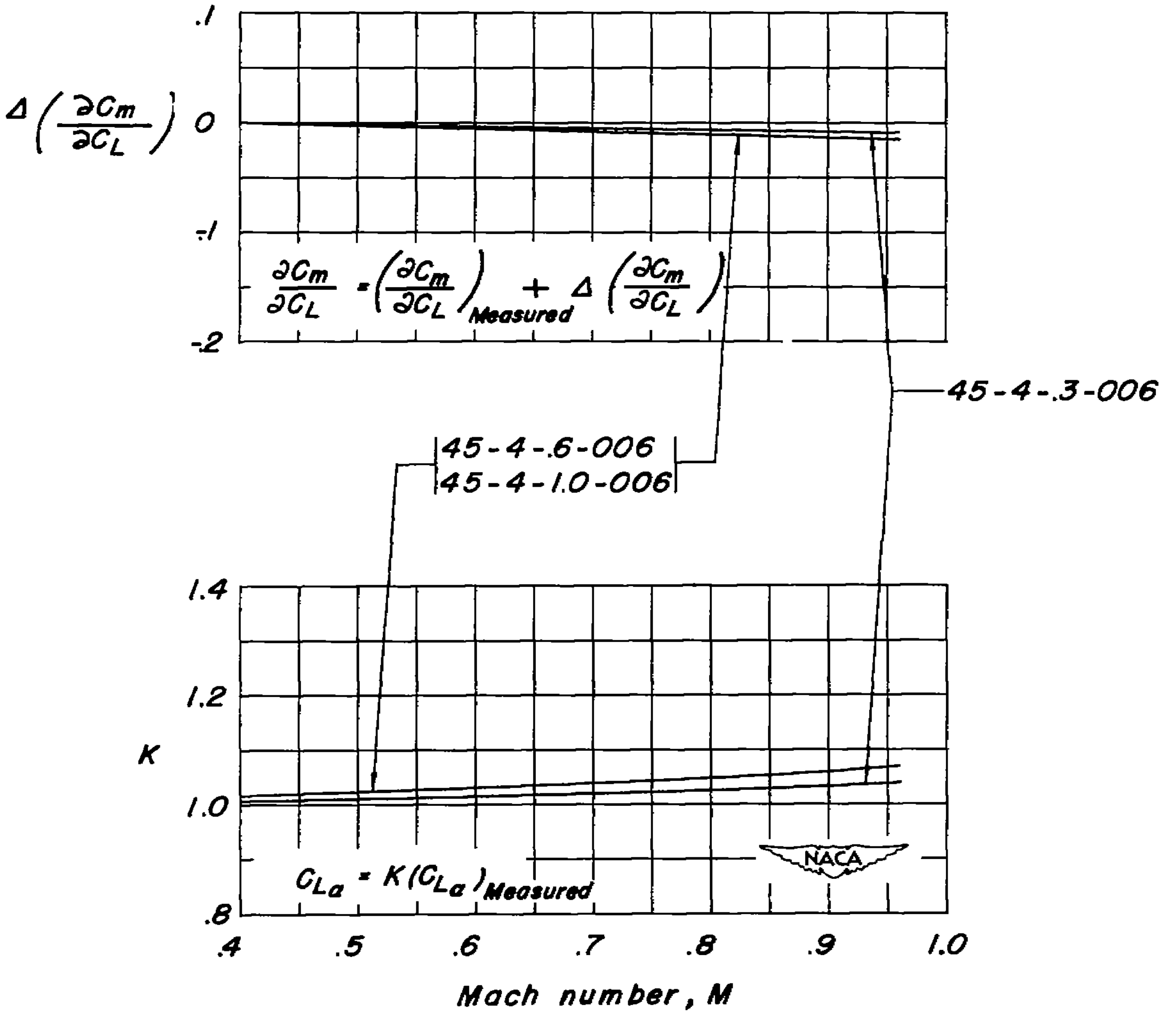


Figure 5.- Correction factors used to correct the summary data for the effects of aeroelastic distortion.

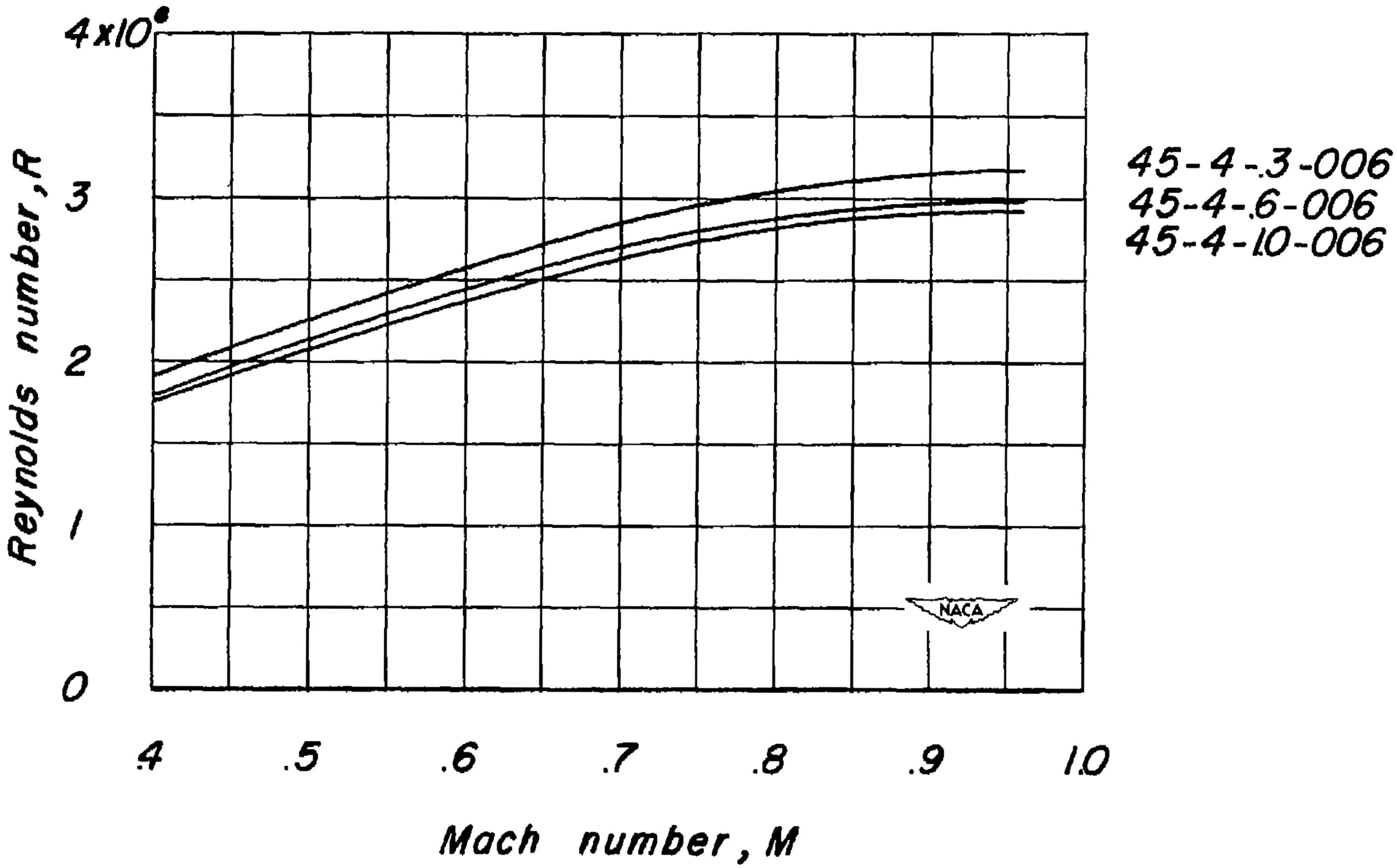
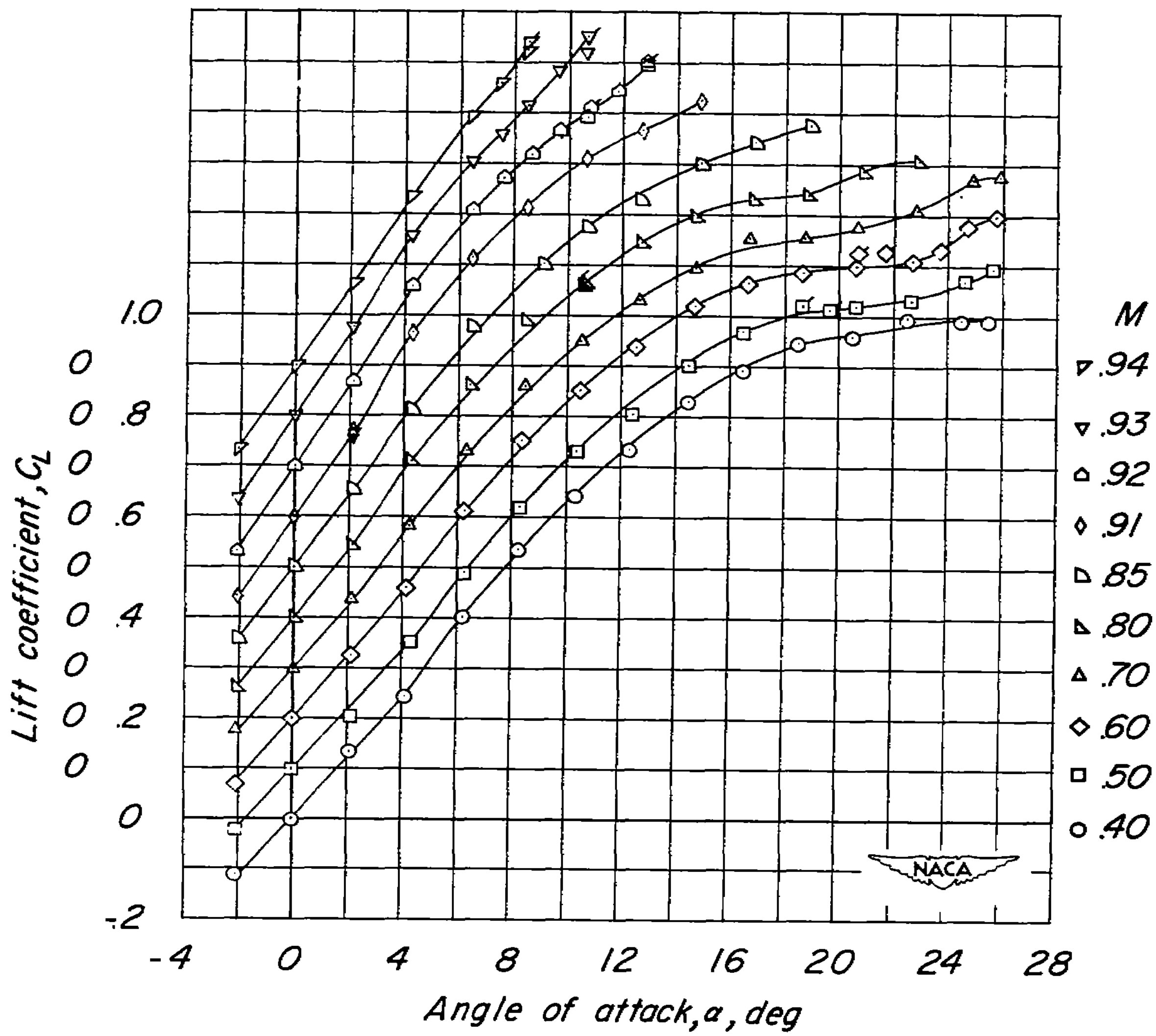
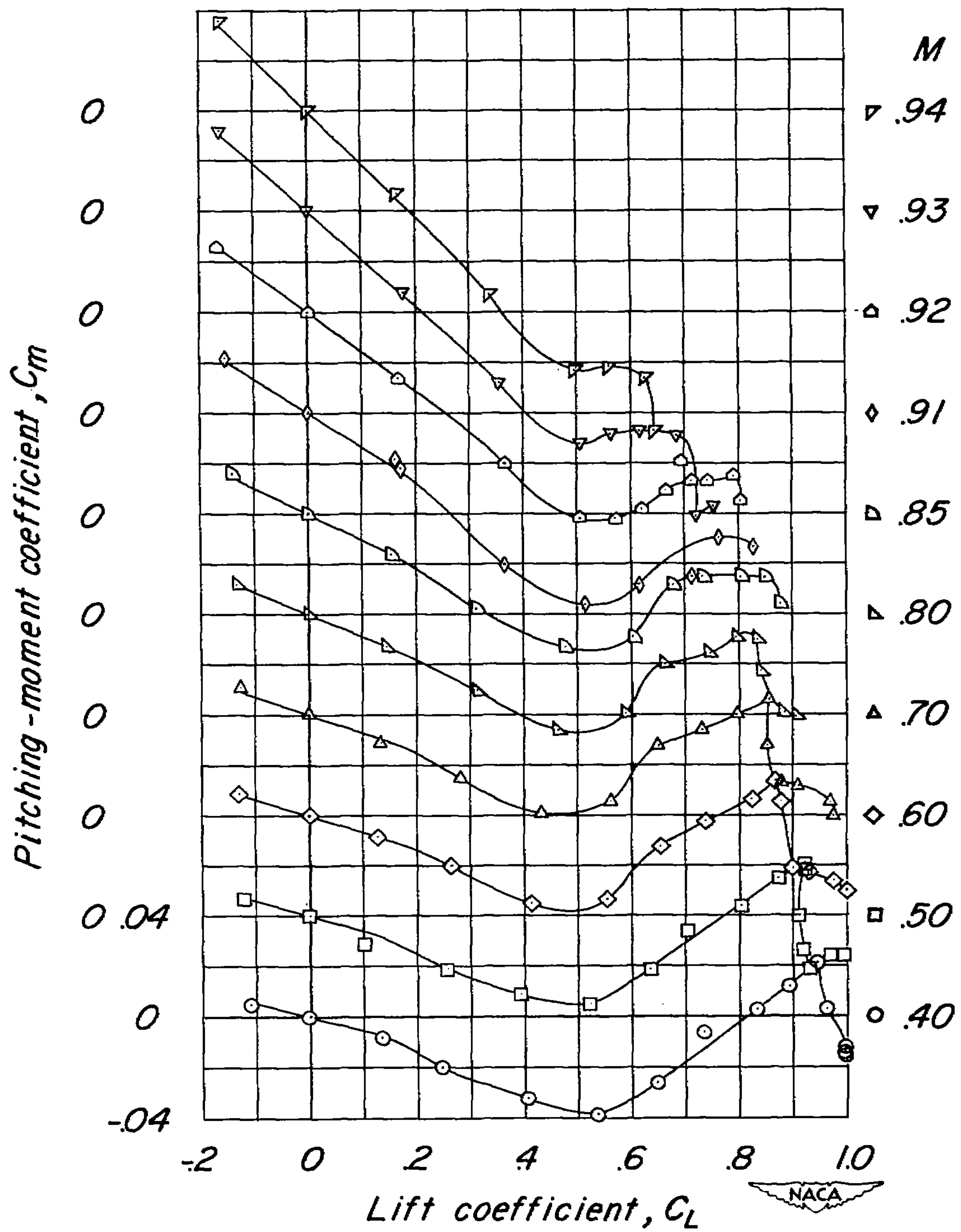


Figure 6.- Variation with Mach number of mean Reynolds number based on the mean aerodynamic chords.



(a) Lift (45-4-0.3-006).

Figure 7.- Aerodynamic characteristics of the taper-ratio-0.3 wing-fuselage configuration. Not corrected for aeroelastic distortion.



(b) Pitching moment (45-4-0.3-006).

Figure 7.- Continued.

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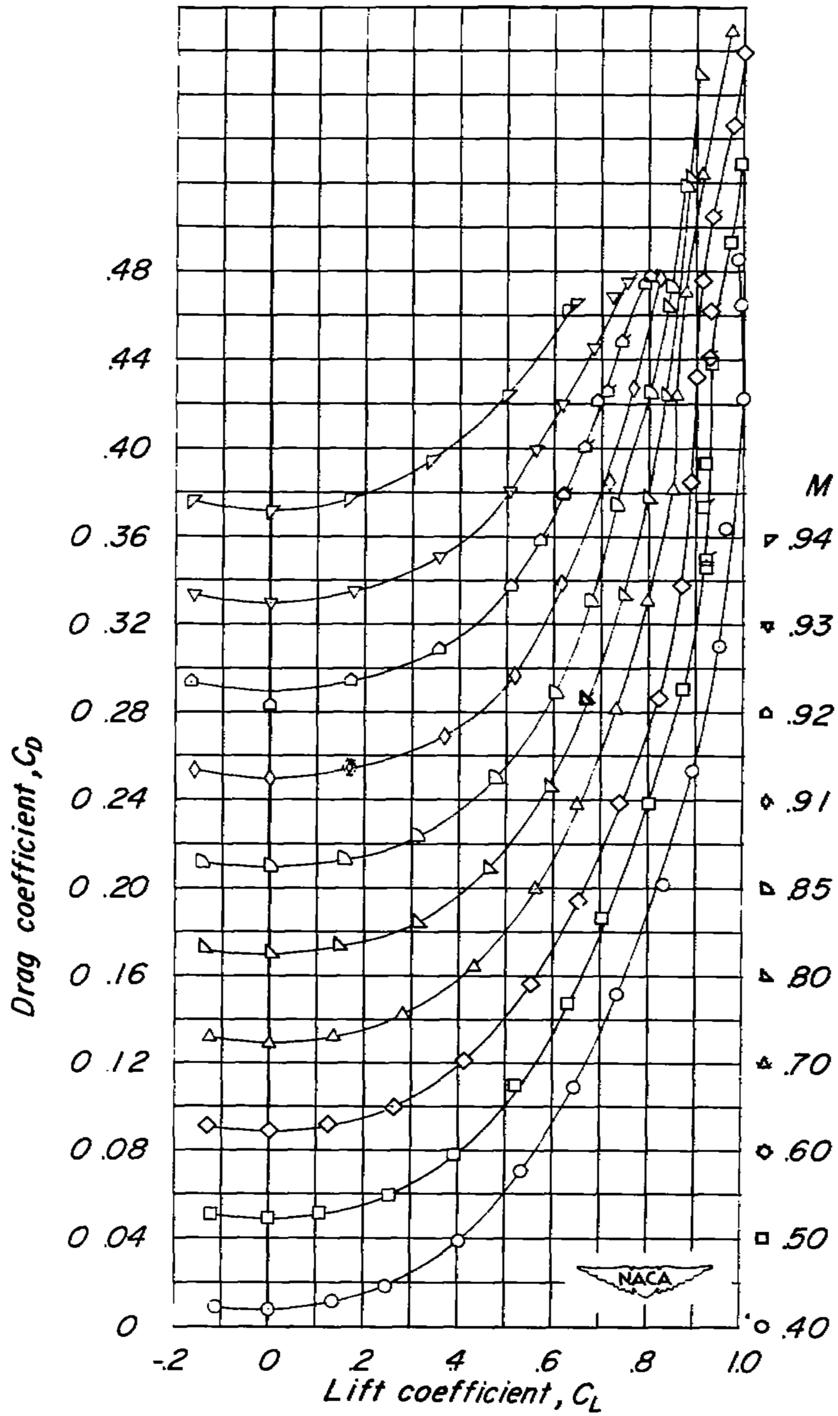
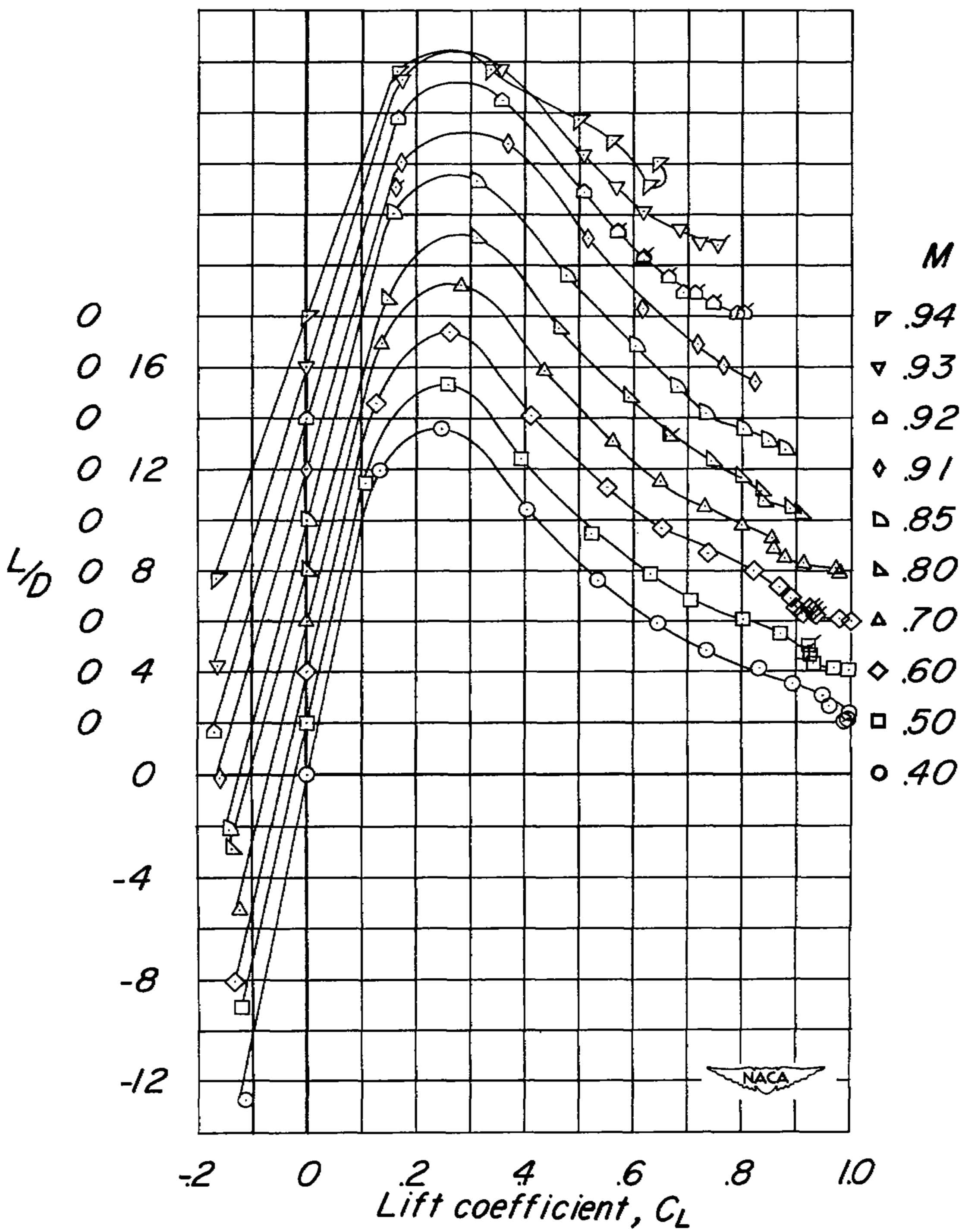


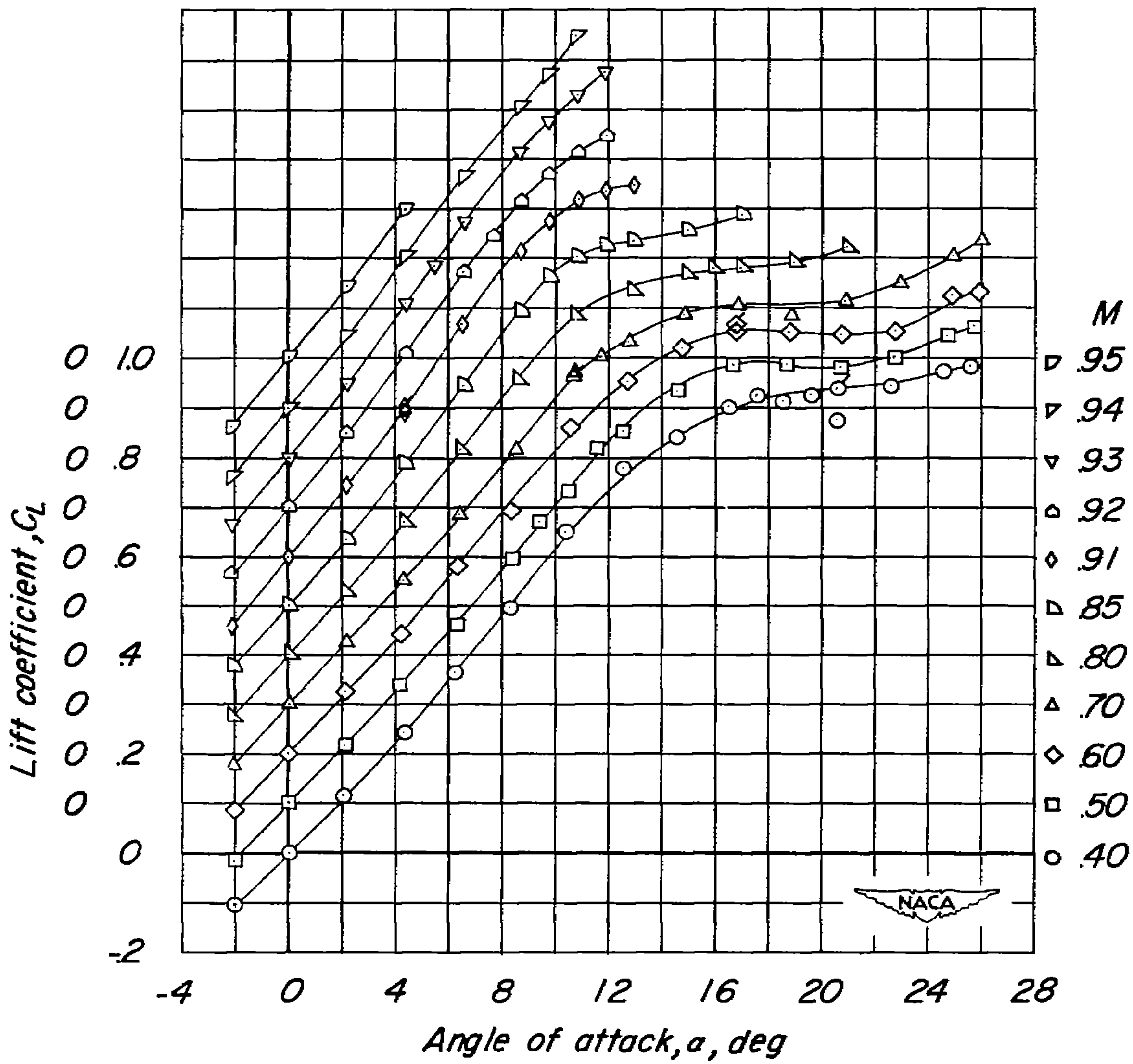
(c) Drag (45-4-0.3-006).

Figure 7.- Continued.



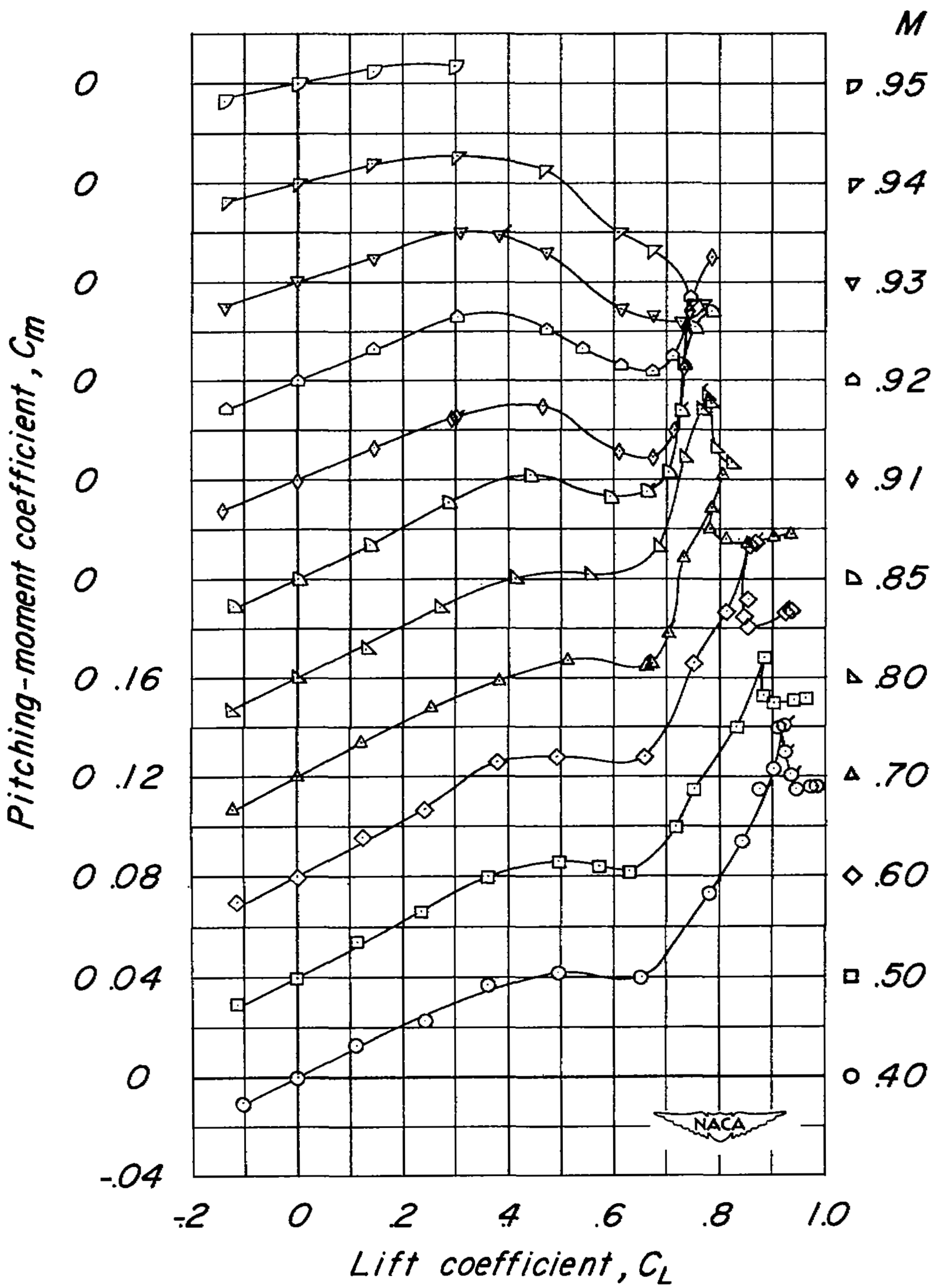
(d) Lift-drag ratios (45-4-0.3-006).

Figure 7.- Concluded.



(a) Lift (45-4-1.0-006).

Figure 8.- Aerodynamic characteristics of the taper-ratio-1.0 wing-fuselage configuration. Not corrected for aeroelastic distortion.



(b) Pitching moment (45-4-1.0-006).

Figure 8.- Continued.

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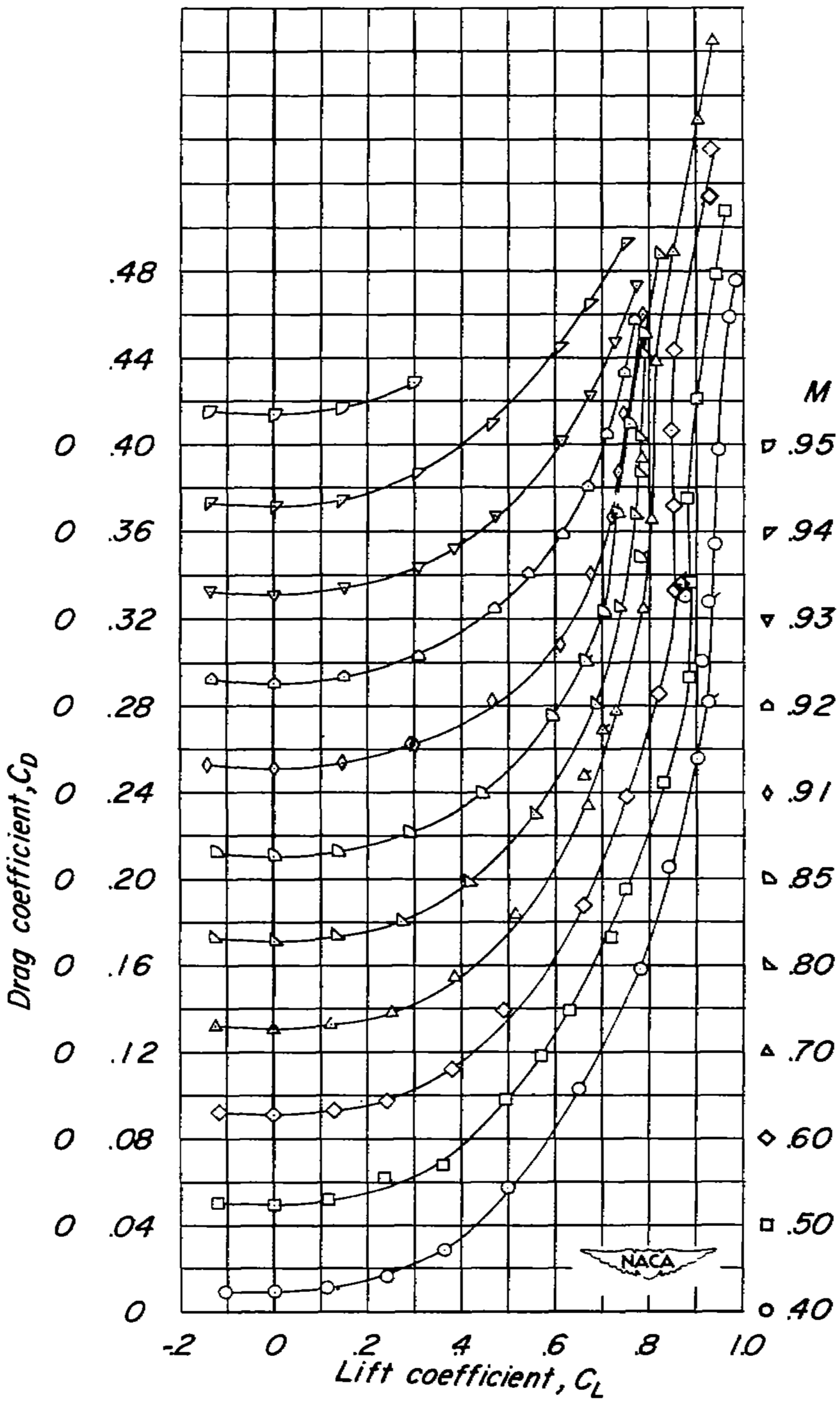
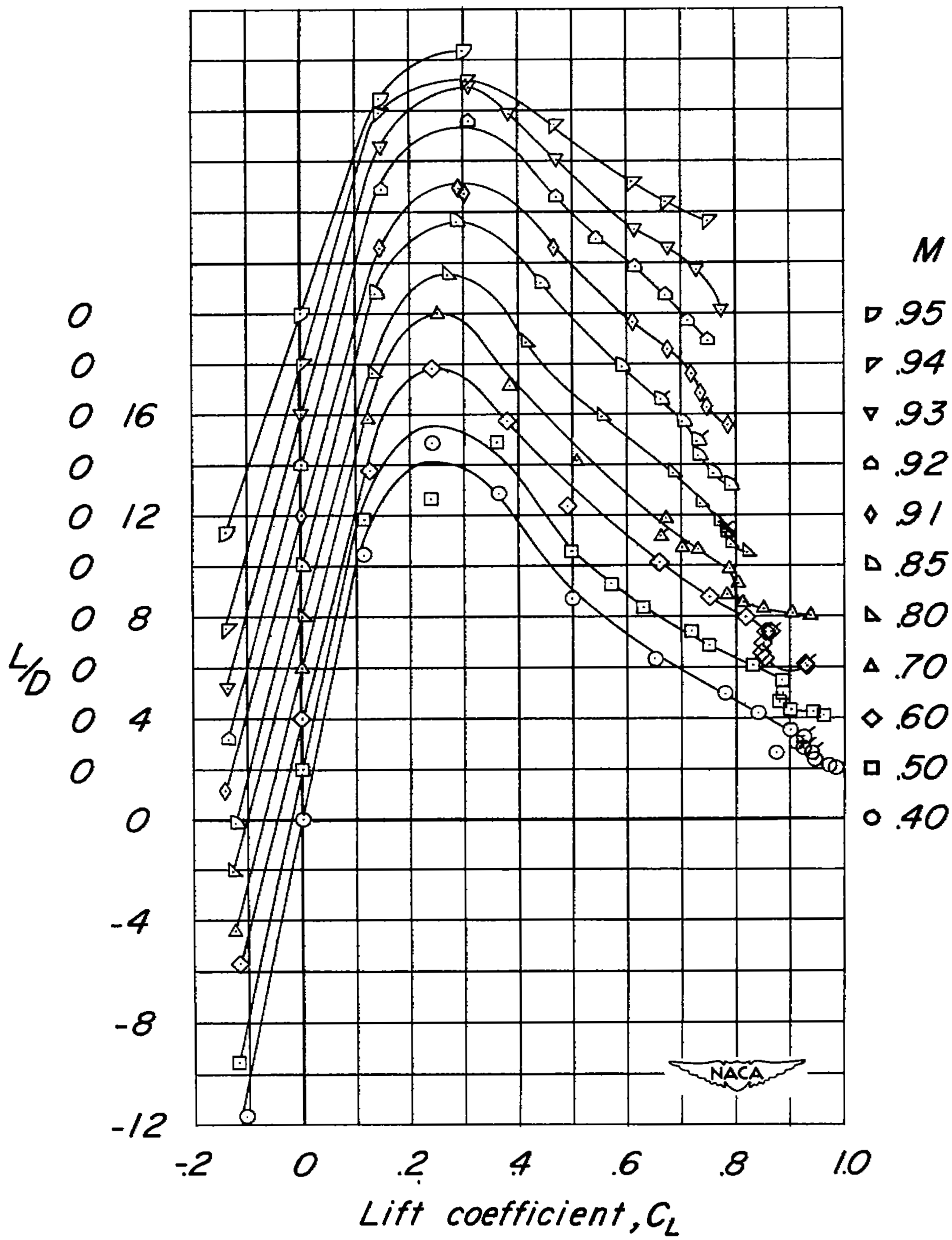


(c) Drag (45-4-1.0-006).

Figure 8.- Continued.



(d) Lift-drag ratios (45-4-1.0-006).

Figure 8.- Concluded.

[redacted]

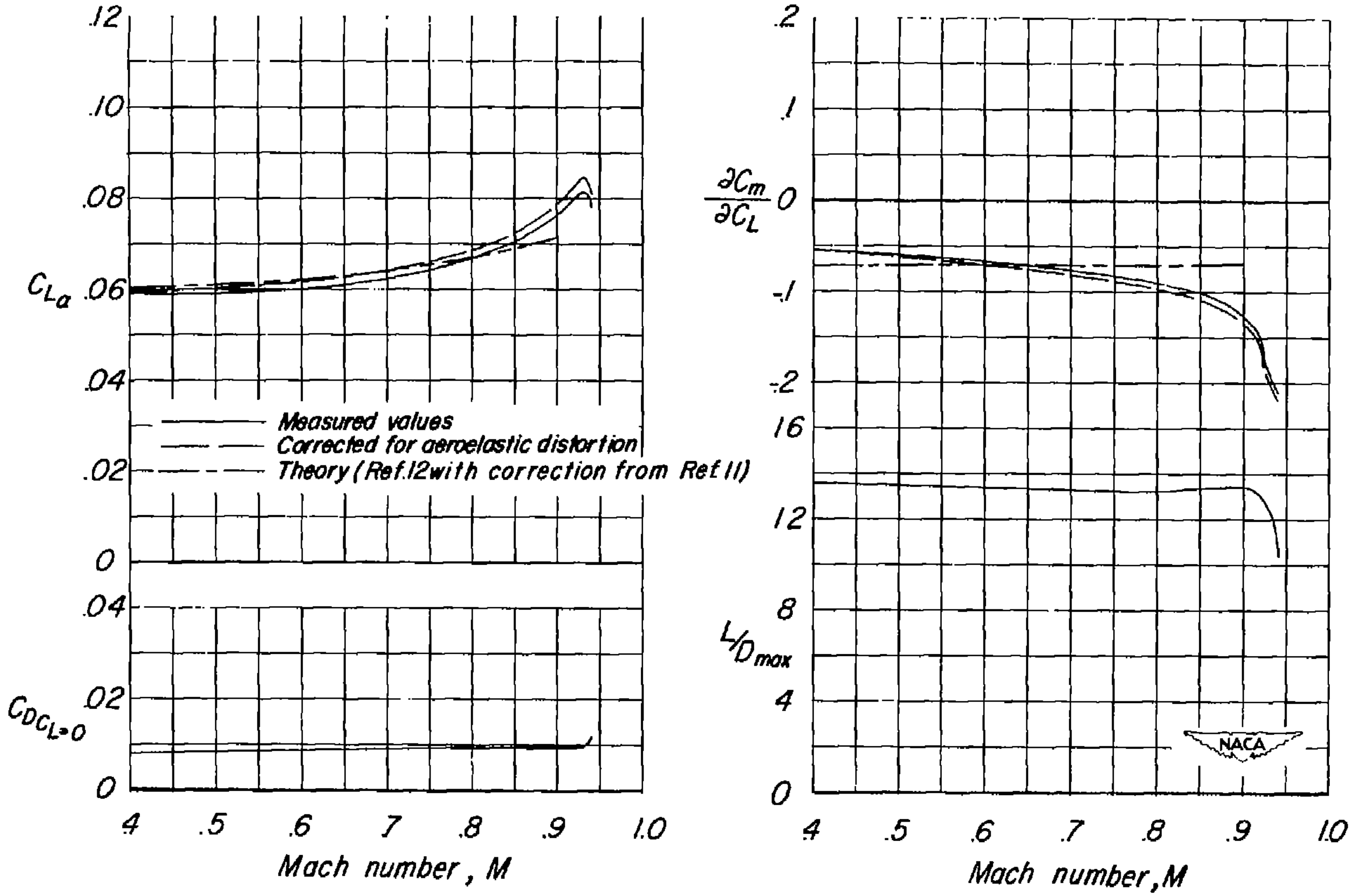


Figure 9.- Summary of the effect of Mach number on the aerodynamic characteristics of the 45-4-0.3-006 wing-fuselage combination.

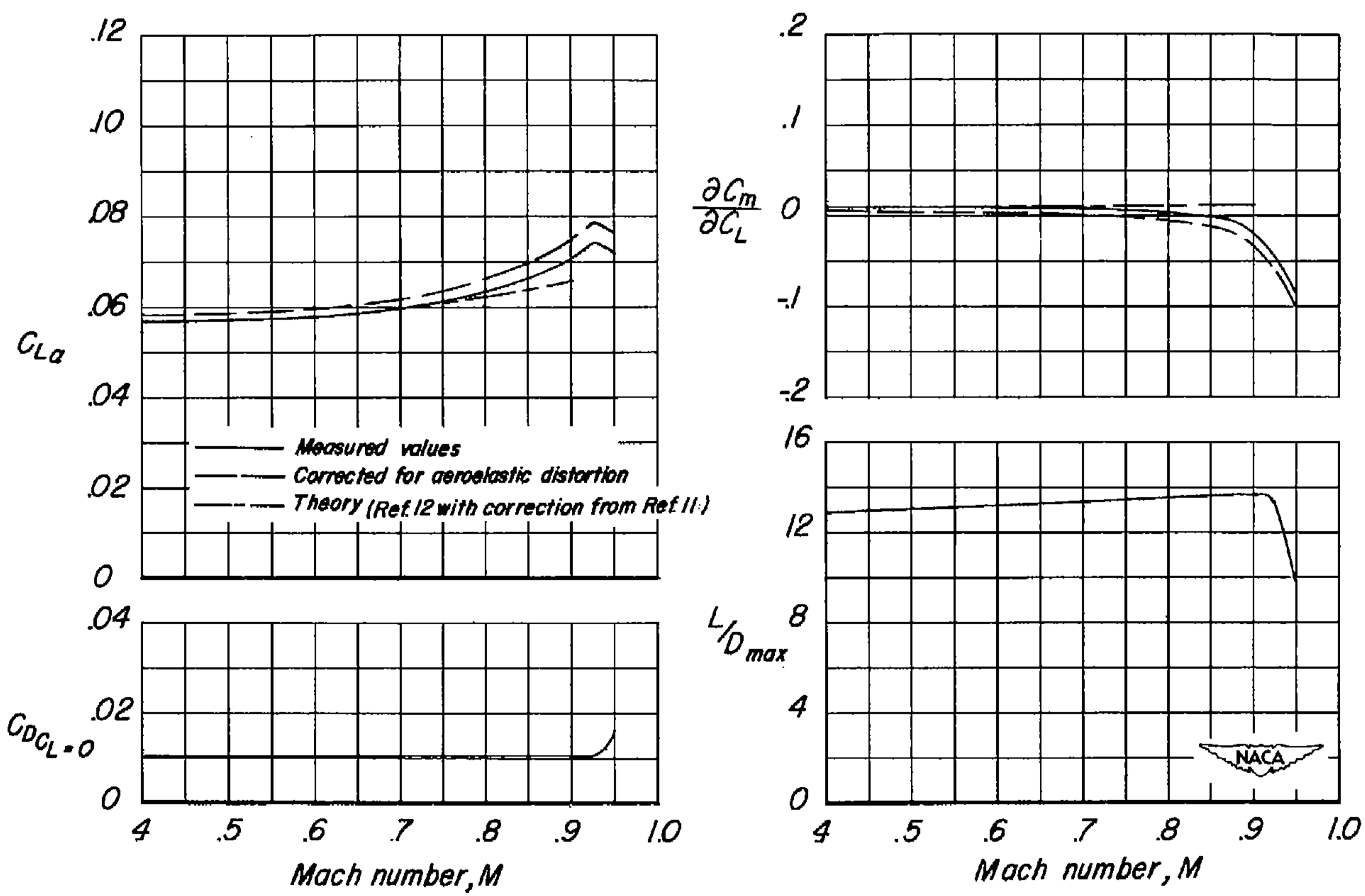


Figure 10.- Summary of the effect of Mach number on the aerodynamic characteristics of the 45-4-0.6-006 wing-fuselage combination.

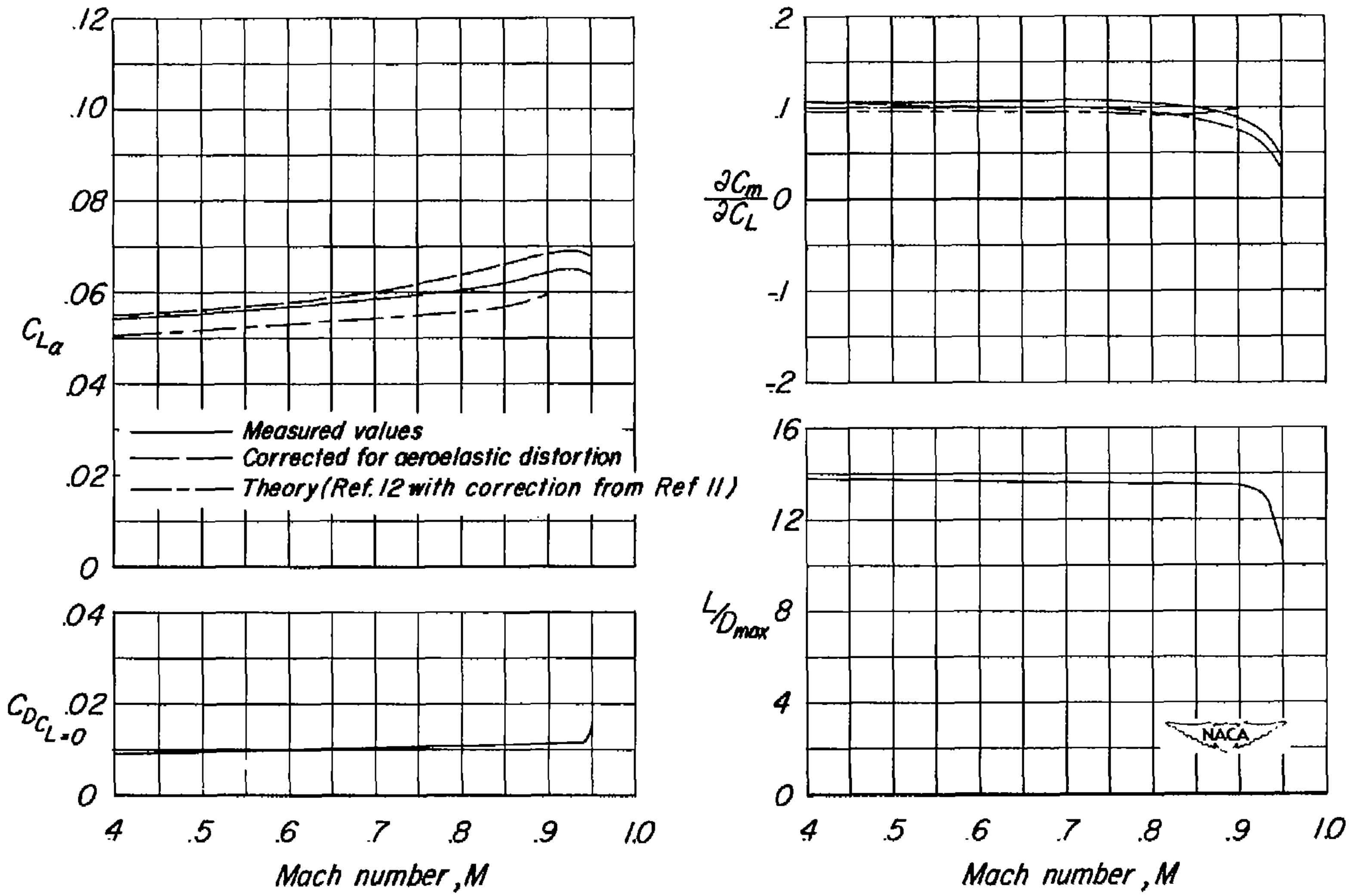


Figure 11.- Summary of the effect of Mach number on the aerodynamic characteristics of the 45-4-1.0-006 wing-fuselage combination.

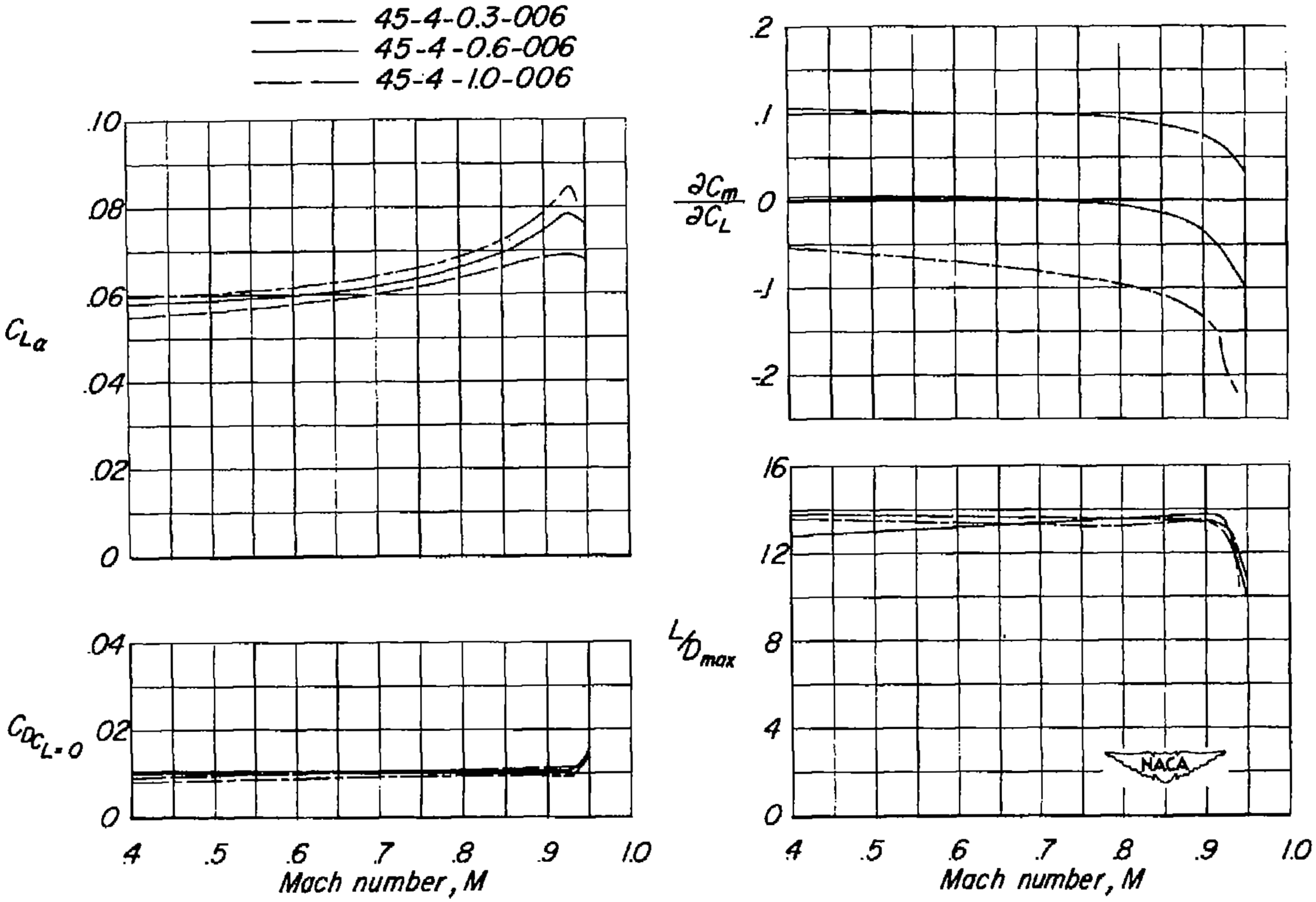


Figure 12.- Comparison of the effects of Mach number on the aerodynamic characteristics of the three wing-fuselage combinations. (C_{L_α} and $\partial C_m/\partial C_L$ corrected for aeroelastic distortion.)

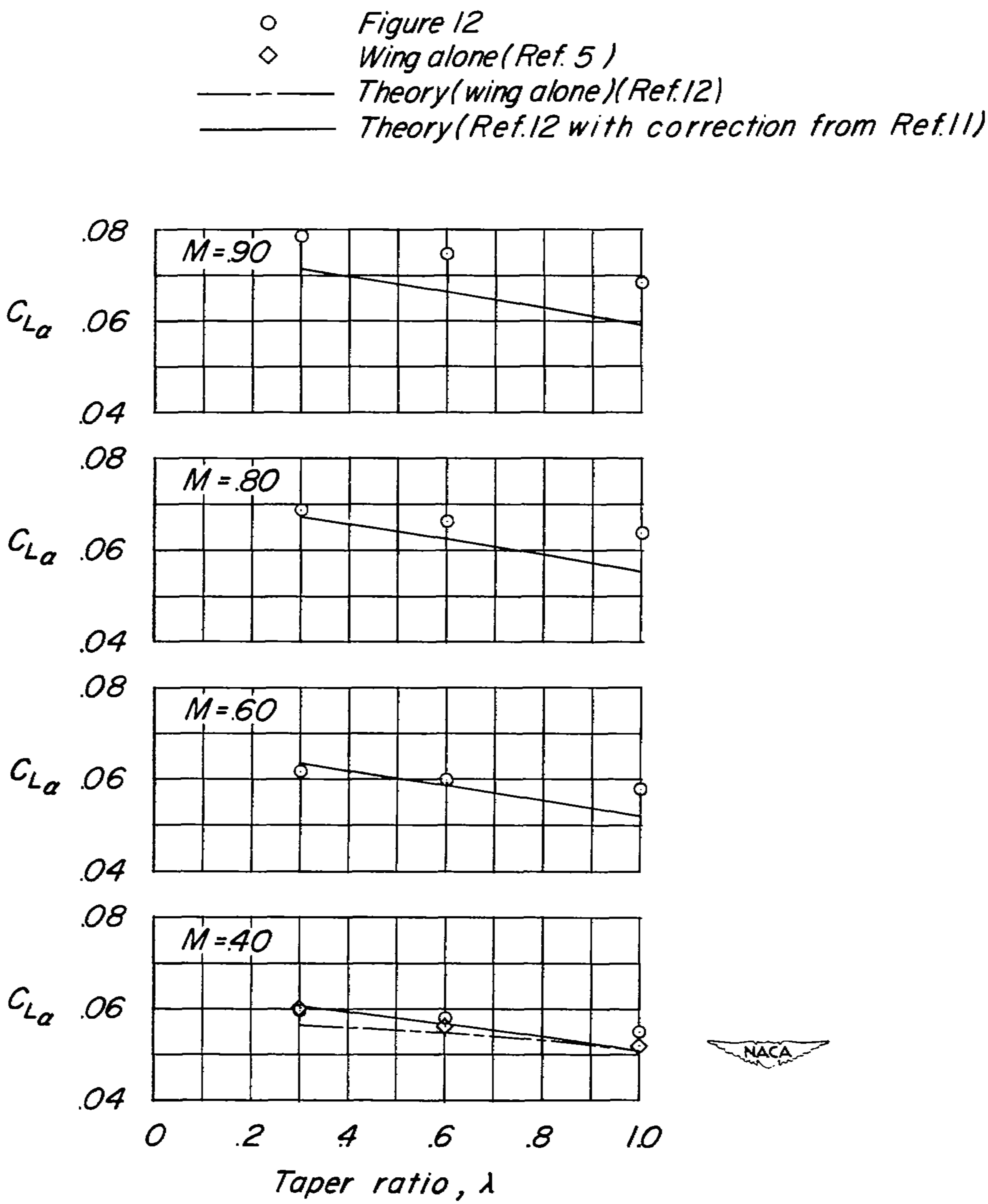


Figure 13.- Effect of taper ratio on the lift-curve slope at four Mach numbers. (Corrected for aeroelastic distortion.)

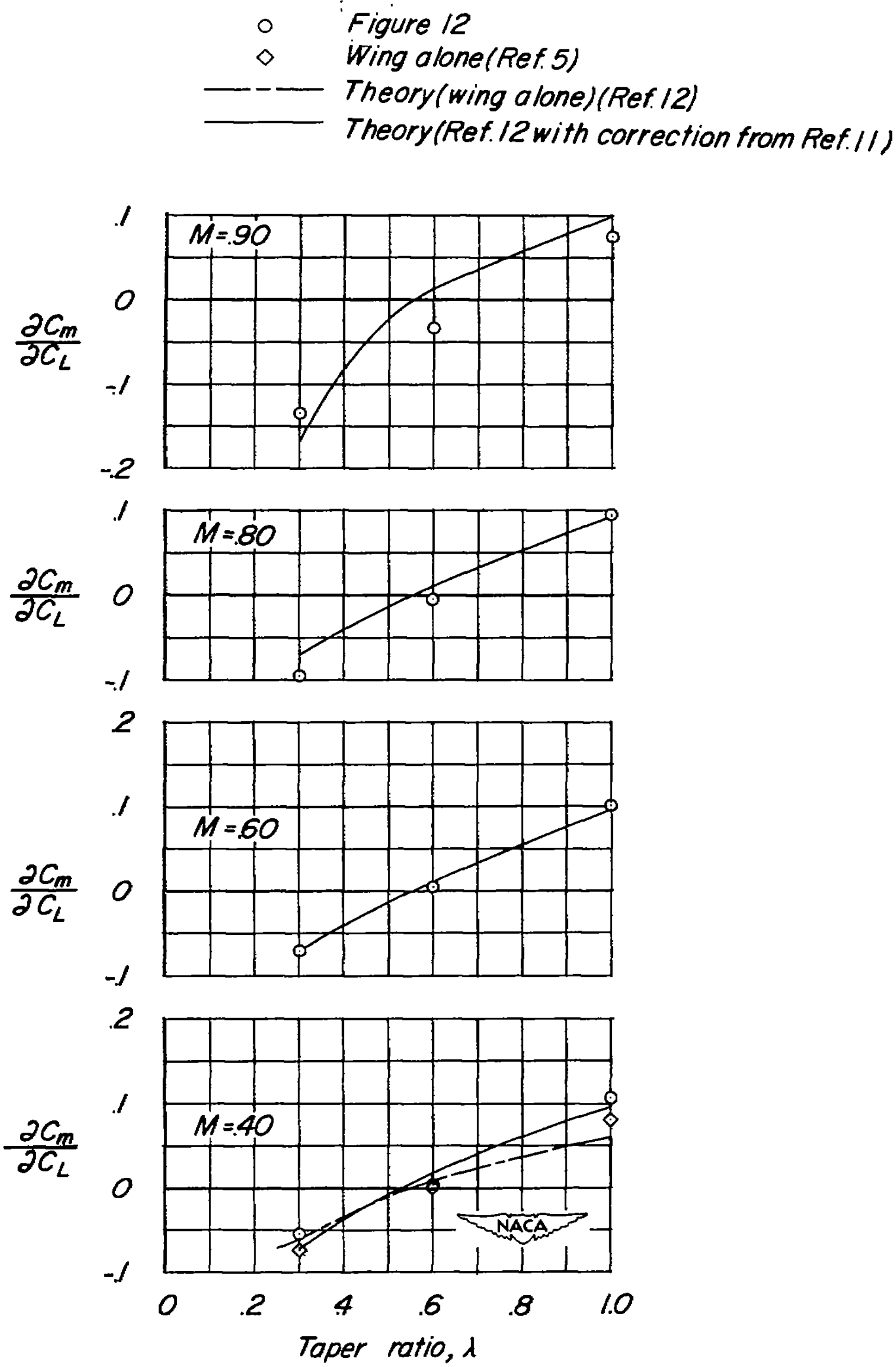


Figure 14.- Effect of taper ratio on the aerodynamic-center location at four Mach numbers. (Corrected for aeroelastic distortion.)

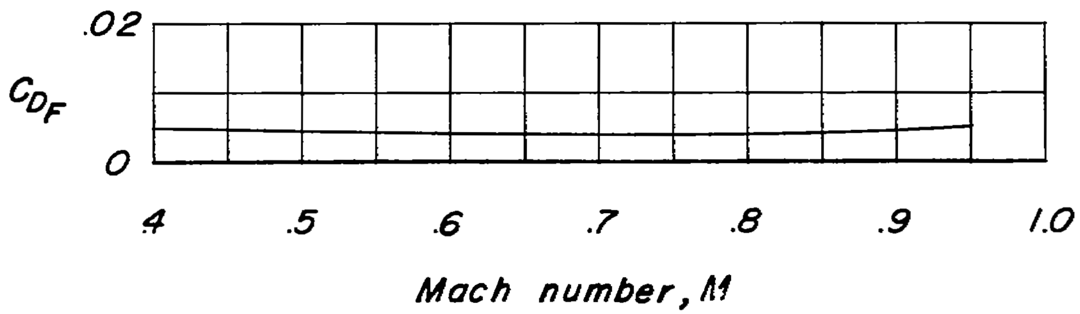


Figure 15.- Drag of the fuselage alone at zero angle of attack, based on the wing area. (Ref. 5.)

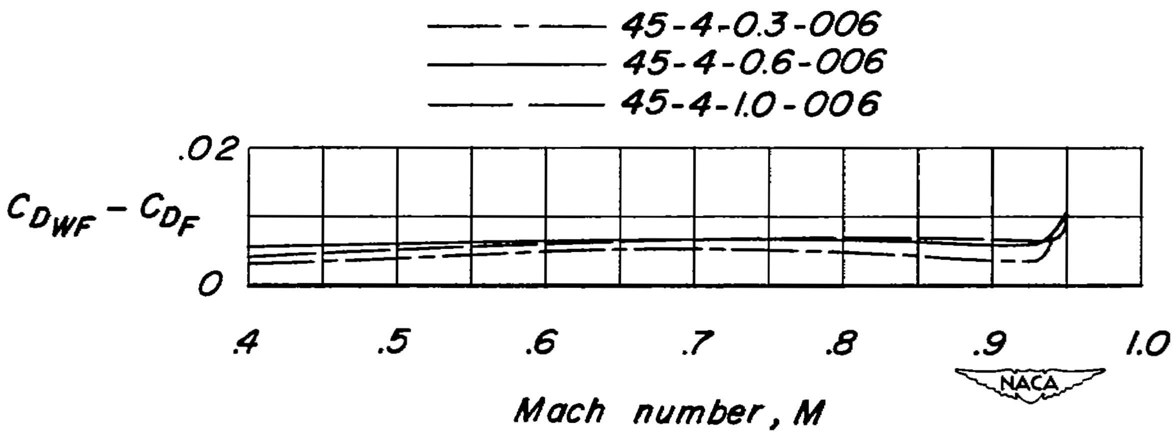


Figure 16.- Variation with Mach number of the wing plus wing-fuselage interference drag at zero lift for the three wings tested.

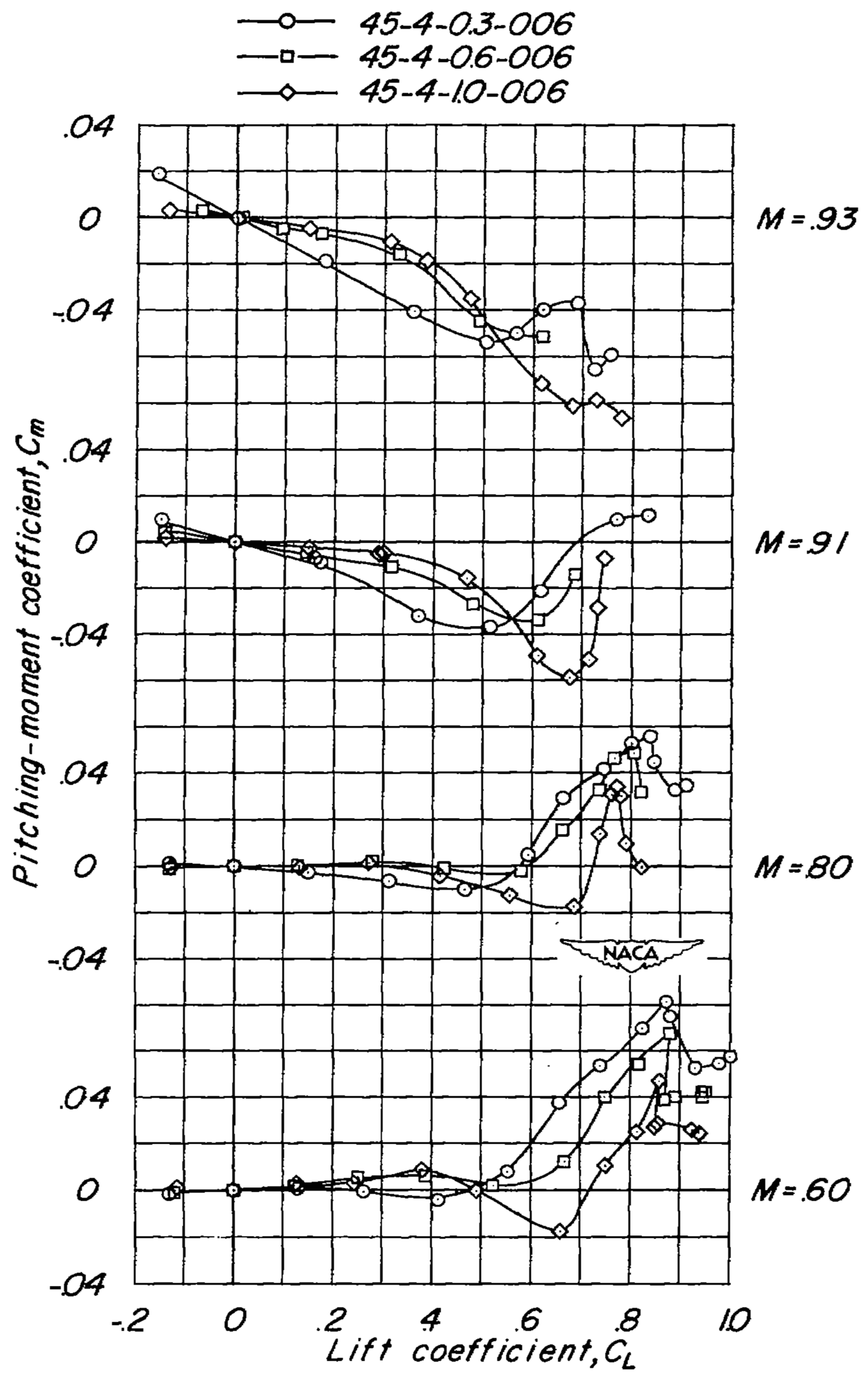


Figure 17.- Effect of taper ratio on the variation of pitching-moment coefficient with lift coefficient. The values of $\partial C_m/\partial C_L$ of the wings of taper ratio 0.3 and 1.0 are adjusted to the same value of the wing of taper ratio 0.6 at $C_L = 0$ and $M = 0.6$.

[redacted]

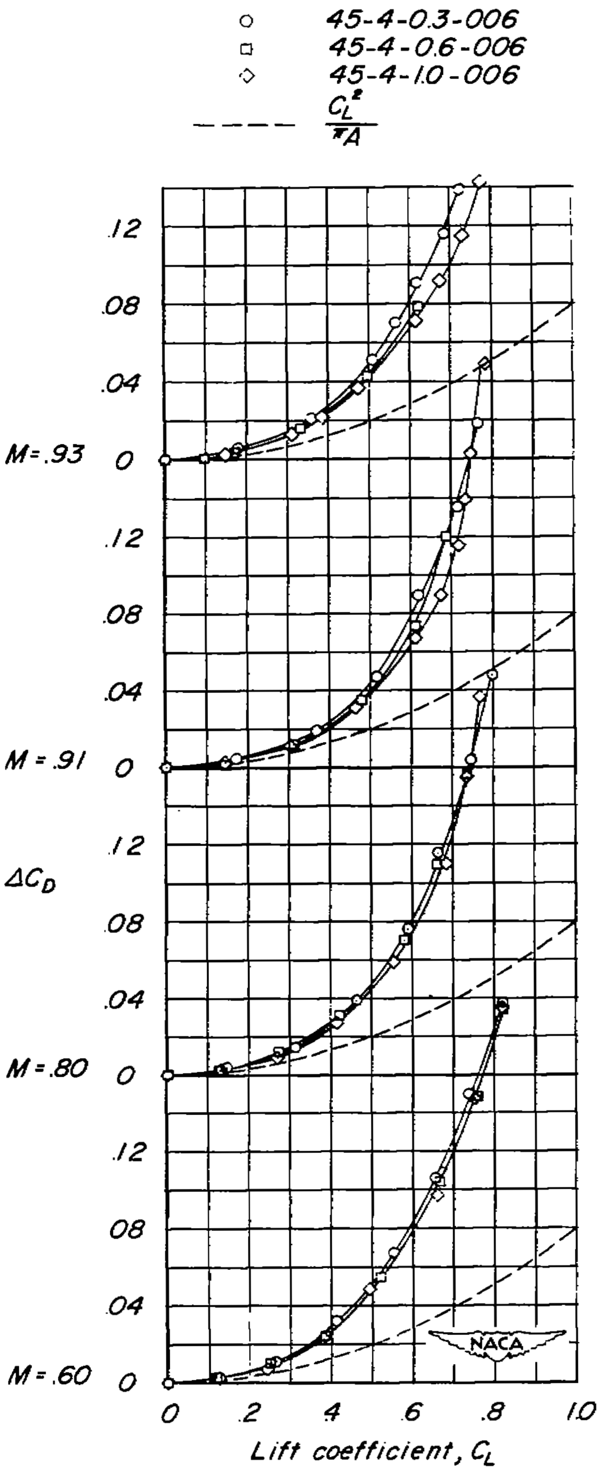


Figure 18.- Comparison of the effects of taper ratio on the drag due to lift at four Mach numbers.

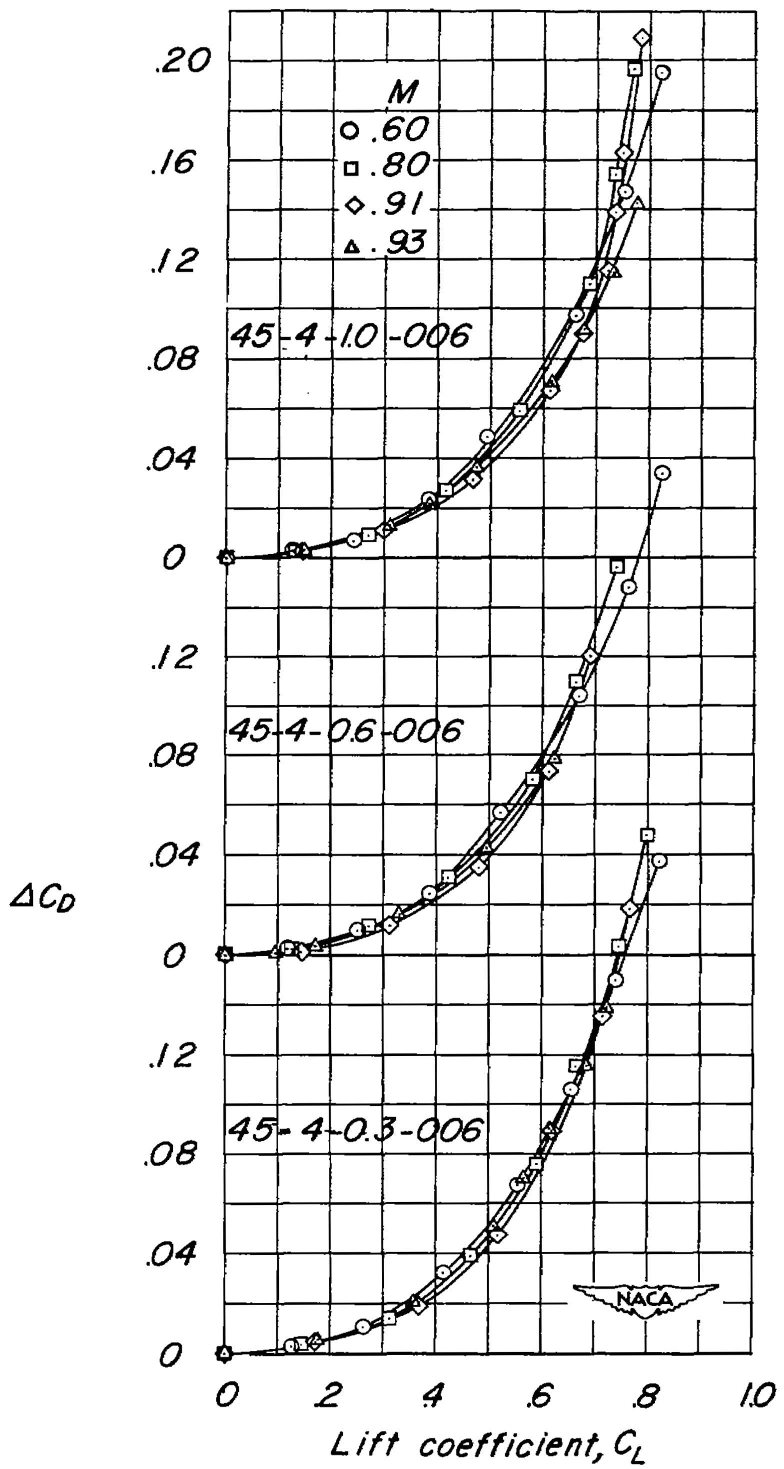


Figure 19.- Comparison of the effects of Mach number on the drag due to lift for the three wing-fuselage combinations.

[redacted]

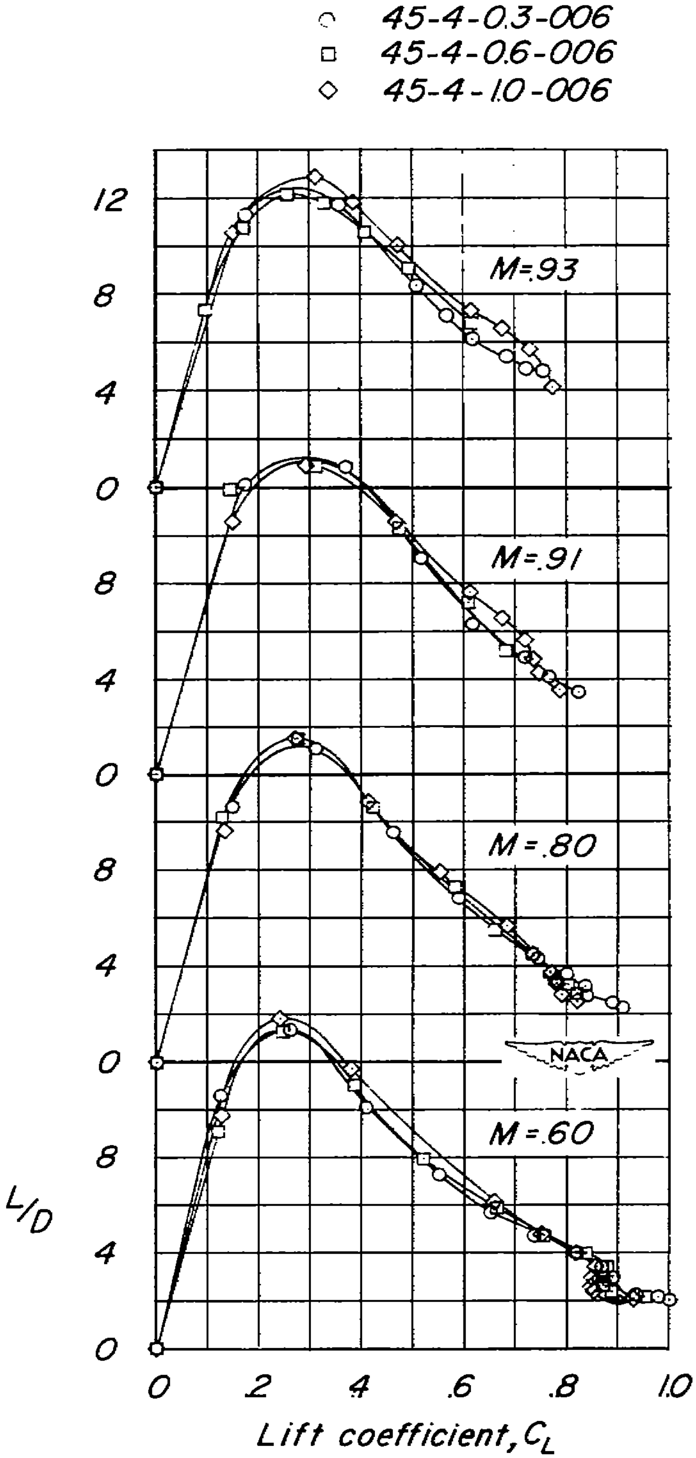


Figure 20.- Comparison of the lift-drag ratios of the three wing-fuselage combinations at four Mach numbers.

SECURITY INFORMATION

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